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THE TRANSONIC AND SUPERSONIC AREA RULE

(A series of lectures given at Rhode-Saint-Genèse, January 1958)

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of the U.S.A.

Rhode-Saint-Genèse, Belgium.

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INTRODUCTION

The area rule is a means for relating the broad shock waves produced by configuration at transonic and supersonic speeds and the resulting wave drag to the cross-sectional areas of the airplane. In the original conception, the area rule was relatively simple. However, throughout the past several years, a number of extensions of the rule have been developed which, while greatly increasing the effectiveness of application of the concept, have resulted in considerably greater complexity. These lectures will be concerned generally with the physical basis, the development, the application, and limitations of the basic rule and its extensions.

The area rule has been utilized to significantly reduce the supersonic drag of a number of American airplanes. The most fruitful application has been with a multi-engine supersonic bomber. The drag of this configuration for the supersonic flight speeds was reduced by as much as one third by the reshaping of the fuselage and by changing the positions of the engine nacelles and external bomb as indicated by the area rule. Because of its basic limitations, it is probable that the area rule cannot be used to improve the drag characteristics of high performance airplanes and missiles flying at Mach numbers greater than roughly a value of 2.0. However, future transport, logistic, and ground-support type airplanes will probably fly in the speed range where the area rule is applicable. Applications of recent extensions of the area rule will probably result in significant improvement of the drag characteristics of such types of airplane configurations.

The lectures will include : first, a discussion of the transonic area rule as interpreted from experimental measurements and theoretical calculations; secondly, a description of the supersonic area rule together with several theoretical extensions of the supersonic area rule to account for the asymmetry of the flow field at supersonic speeds; thirdly, discussions of certain refinements required in practical application, the effects of airplane design parameters on the effectiveness of area-rule modifications, and limitations in the applications, and fourthly, a description of methods for utilization of the area rule in delaying the onset of drag rise at high subsonic speeds.

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TRANSONIC AREA RULE

Basic Rule

Following the completion of the 8-foot transonic tunnel, comprehensive research programs were carried out in this new facility to determine the effects of a wide range of variables on wing-fuselage interference at transonic speeds and to study the flow over a typical configuration at these speeds. The results of the initial systematic program indicated large, highly variable wing-fuselage interference at these transonic speeds. The flow surveys showed shock patterns completely independent of the wing geometry. Consideration of these results, together with an analysis of the general physical nature of the flow at transonic speeds, has led to the conclusion that near the speed of sound the drag rise for a thin low-aspect-ratio wing-body combination is primarily dependent on the axial distribution of cross-sectional area normal to the airstream (Reference 1). (The drag rise, sometimes referred to as pressure drag, is the difference between the drag level near the speed of sound and the drag level at subsonic speeds where the drag is due primarily to skin friction.) In order to illustrate the concept, figure 1 shows a wing-body combination and a body of revolution. A typical cross section normal to the airstream for the wing-body combination is shown at AA. The cross-sectional area of the wing is wrapped around the body of revolution so that the body has the same cross-sectional area at BB. All the other cross-sectional areas of the body of revolution are the same as those for the wing-body combination at the same axial stations. On the basis of the conclusion just stated, the drag rise for this body of revolution should be similar to that for the wing-body combination.

This relationship of the drag-rise increments for the wing-body combination and the comparable body of revolution is due primarily to the general similarities of the major portions of the extensive flow fields of the configurations. These similarities are illustrated in figure 2 which presents schlieren photographs of the flow fields for an unswept wing-body combination, together with that for an equivalent body of revolution. The combination has been rolled to three positions so that side, plan, and intermediate views are seen. Near the edges of the pictures, the observed shocks for the combination in each view are generally similar to those for the equivalent body. These comparisons are indicative of the similarities of the extensive fields beyond the view of the schlieren. Near the configurations there are some differences of the flow fields for the wing-body combinations and equivalent bodies of revolution. However, the major portion of the energy losses associated with the shocks is produced in the extensive regions at appreciable distance from the configuration. Therefore, from a drag standpoint, it may be assumed that these differences near the configuration are of secondary importance. The general similarities of the extensive flow fields at distances from the configuration may be attributed to several aerodynamic phenomena characteristic of flow near the speed of sound. First, the field of any given displacement is concentrated in a plane nearly normal to the airstream. Because of this fact, the streamwise locations of the effects of the displacements of the wing are essentially the same as those for the corresponding effects produced by the comparable body of revolution. Secondly, at these considerable lateral distances from the configuration, the field is primarily dependent on the general displacement of the configuration rather than on the details of the shape.

The generally close similarities of the effective fields for the wing-body combination and the comparable body of revolution in the regions producing the main portion of the shock losses suggests that the energy losses associated with the shocks for the two configurations should be similar. Since the drag rise for thin low-aspect-ratio wings is due primarily to shock losses, the drag rise for the combination should be approximately the same as that for the equivalent body of revolution.

In figure 3, the measured drag-rise increments for various swept-, delta-, and unswept-wing-body combinations and complete airplanes at a Mach number of 1.03 are compared with the increments for equivalent bodies of revolution. The aspect ratios of the wings are 4 or less and the thickness ratios are 7 percent or less. Except for one configuration, there is a general qualitative agreement between these drag-rise increments. Deviations from exact agreement are due to second-order effects, such as differences of the flow fields as shown in figure 2. The single case of marked disagreement is for a swept wing airplane configuration. This disagreement is associated with the small positive trailing edge sweep angle for this configuration. As would be expected, the correlation between the drag-rise increments of the wing-body combinations and the equivalent body of revolution generally becomes less close as the Mach number is increased beyond 1.0. The severity of this divergence varies markedly depending on the configuration.

Theory

Up to this point, we have discussed the experimental verification of the area rule. The theoretical work related to the area rule will now be discussed. Wallace D. Hayes in his 1946 thesis on Linearized Supersonic Flow (Reference 2) developed equations for computing supersonic wave drag by considering the effect of source and sinks distributed over the surface of the body on the flow field at a distance from the body. His equations indicated that when the Mach number approaches 1.0, the circumferential variations of the effects of the configuration on the flow field at a distance tend to disappear. However, because of the limitations of the theory at transonic speeds, this result was not thought to be of practical significance. Later, G.N. Ward (Reference 3), E.W. Graham (Reference 4), and K. Oswatitsch (Reference 5), and others restricted themselves to very narrow or slender shapes and expressed the wave drag in terms of the longitudinal area distribution for Mach numbers above 1.0 where the linear theory has a better justification. More recently, the theory for the Mach number of 1.0 area rule has been greatly extended by a number of people. Sune B. Berndt of the Aeronautical Research Institute of Sweden (Reference 6), W.T. Lord, Royal Aircraft Establishment, Keith C. Harder and E.B. Klunker of the Langley Research Center of the National Advisory Committee for Aeronautics (Reference 7) and a number of people at the Ames Research Center of the NACA have contributed to the theory.

Since the area rule relates the drag of wing-body combinations to equivalent bodies of revolution, it might be expected that the drag of a wing-body combination could be computed utilizing linear theories for bodies of revolution. However, it has been shown experimentally by a number of experiments that this linear theory does not predict accurately the drag of a body of revolution near the speed of sound. Theory predicts an abrupt increase in drag by the body of revolution at supersonic speeds while experiments generally indicate a gradual increase in the wave drag for a body of revolution from Mach numbers somewhat below the speed of sound to slightly above the speed of sound. The drag at the speed of sound is normally half of the fully developed wave drag at slightly supersonic speeds. This gradual build up of the wave drag is associated with the gradual transition of the flow conditions from the usual subsonic phenomena to the conditions assumed in the theory; that is, a condition with the bow shock essentially attached to the nose of the bodies and the aft shock essentially at the base of the bodies. Because of this development, the agreement between drag determined on the basis of experimental body theory and experimental drag near the speed of sound has been poor. It is now generally accepted that computations of the wave drag must be limited to Mach numbers above approximately 1.05. The computation of wave drag for low supersonic Mach numbers will be discussed in the following lectures.

Fuselage Shaping

It would be expected on the basis of this concept that, near the speed of sound, the minimum drag rise would be obtained by designing a wing-body combination with an area distribution similar to that for a smooth body of revolution with the highest possible fineness ratio. The fineness ratio that should be used is probably considerably less than that required for minimum total drag because of such problems as airplane stability and structural weight. One method of obtaining this favorable area distribution is to reshape the body. A number of experiments have been made to determine the effectiveness of such reshaping. Representative results, obtained in the Langley 8-foot transonic tunnel, are presented in figure 4.

On the left-hand side of this figure are shown the effects of such a body modification on the zero-lift drag-rise characteristics of a 6-percent-thick, aspect ratio 4, 45° swept-wing-body combination. The solid line shows the variation of drag for the wing in combination with a body of revolution of fineness ratio of 11. The wing is placed on the body in such a manner that the leading edge of the wing is at the maximum diameter of the body. With this arrangement, the indentation used did not change the maximum cross-sectional area of the body. The dashed lines are the results obtained for the wing in combination with a body of revolution indented circularly to obtain the same area distribution as for the original body alone. For comparison, the results for the body alone are also shown. Indentation eliminated approximately 90 percent of the drag rise associated with the wing at Mach numbers from 1.00 to 1.05. The difference from that expected may be attributed to secondary flows not accounted for by this linear theory. When the Mach number is increased beyond 1.05, the drag rise for the indented wing-body combination approaches that for the original wing-body combination.

On the right-hand side of figure 4 are presented the effects of body indentation on the zero-lift drag-rise characteristics for a 4-percent-thick, 60° delta-wing-body combination. The solid curve shows the drag characteristics for the wing in combination with a body of revolution having a fineness ratio of 7.5. The dashed line indicates the drag variation after the body has been indented circularly to produce an area distribution for the combination the same as that for the original body alone. In this case, the indentation reduced the maximum cross-sectional area of the body somewhat. It may be noted that again a significant reduction in the drag rise was obtained by such an indentation at transonic speeds. However, in this case, the drag rise for the indented wing-body combination is significantly greater than that for the body alone. This deviation from the result which might be expected on the basis of the area-distribution concept is probably due primarily to increased secondary flows associated with the straight wing trailing edge and the abrupt contours of the fuselage.

Results obtained with smooth-surfaced configurations have indicated a marked reduction in drag at subsonic speeds associated with the use of indentation with swept and delta wings. However, with fixed transition this difference is not present. The influence of surface conditions on the effects of indentation apparently decreases with increase in the Mach number to supersonic speeds. Obviously, the volume of the indented wing-body combination is not as great as that for the original wing-body combination. However, increasing the size of the body to recover the volume lost in indentation would increase the drag for the indented combination by a small fraction of this reduction in drag obtained.

The question now might arise as to whether it would be possible to obtain drag reductions at transonic speeds by adding to an existing wing-body combination to obtain a more favorable area distribution. Recently, investigations have been made of such additions on a 60° delta-wing airplane. Results are presented in figure 5. First, the fuselage was extended approximately 8 percent to obtain a more favorable area distribution of the rearward portion of the airplane. This addition resulted in significant reductions in the drag rise. Further reduction was obtained by adding side fairings to the extended configuration to fill the dip in the area distribution as shown. The body lines with these additions were still relatively smooth. Additions which lead to severely irregular body lines would not be recommended.

Effect of Nacelle Position

The influence of external nacelles on the transonic drag rise may also be interpreted on the basis of the area-rule concept. This is illustrated by the experimental results presented in figure 6. Symmetrically mounted nacelles were investigated at a number of spanwise locations on a 45° swept wing with an aspect ratio of 6 and a thickness ratio of 6 percent by the Pilotless Aircraft Research Division. The drag-rise characteristics are shown on the left of the figure; the axial developments of cross-sectional area are shown on the right. The highest transonic drag was obtained with the nacelle position giving the highest maximum area and greatest rates of change of area. The drag rise is progressively reduced with improvements in the area development. For the tip location of the nacelle, a favorable effect on drag was measured. This effect may be attributed to the lengthening of the total area development with the associated reduction of the rates of change of area along the aft portion of the configuration.

SUPERSONIC AREA RULE

Basic Rule

At supersonic speeds, the disturbance fields are conical rather than planar as at transonic speeds and it would be expected that an area-rule design procedure based on a consideration of such conical fields would provide improved drag characteristics at these higher speeds. Such a method, referred to as the supersonic area rule, has been developed on the basis of the theoretical work of W.D. Hayes (ref. 2) and by considering the physical nature of the flow. Most of the redesigns of American airplanes based on the area rule have utilized this extension. As in the previous section, let us consider the physical nature of the flow first, and then consider the theoretical aspects of the problem.

As at transonic speeds, the major part of the wave drag for a wing-body combination at supersonic speeds results from losses associated with shocks at considerable distances from the configuration. Thus the wave drag may be estimated by considering the stream disturbances produced by a configuration at these distances. At moderate supersonic speeds, these disturbances may be considered in individual stream tubes, such as A in figure 7. If small induced velocities are assumed, the effects of changes in the configuration arrive at points on this tube along Mach lines which lie on cone segments, such as B. For reasonable distances from the configuration, roughly 2 spans or greater, and normal, relatively low-aspect-ratio wings, the surface of these cone segments in the region of the configuration may be assumed to be the Mach planes, such as C, tangent to the cone segments between the tube A and the axis of symmetry.

Consideration of the propagation of the local effects of the configuration indicates that the variations in the disturbances at the stream tube A generally may be assumed to be approximately proportional to streamwise changes in the normal components of the total areas of the cross sections, such as DD, intersected by these Mach planes. It follows that the wave losses in the stream tube are functions of the axial distribution of these cross-sectional areas. Obviously, the losses in the set of stream tubes along a given radial sector are functions of one axial distribution of cross-sectional area while those in tubes in circumferentially displaced sectors are functions of various distributions determined by sets of Mach planes with axes of tilt rotated about the axis of symmetry.

It follows from the foregoing considerations that the zero-lift wave drag for a wing-body combination at a given moderate supersonic Mach number is related to a number of distributions of the normal components of cross-sectional areas as intersected by Mach planes which are inclined to the stream at the Mach angle m (figure 8). The various distributions are obtained with the axis of tilt of these Mach planes rolled to various positions around the center line of the configuration. This procedure is illustrated in figure 8. For clarity, the position of the axis of tilt of the Mach plane is maintained and the configuration is rolled. For configurations symmetrical about horizontal and vertical planes, the area distributions are determined for various roll angle θ from 0° to 90°. The approximate wave drag for the combination is an average of functions of a number of area distributions so determined. The area distributions obtained for the configuration shown in figure 8 with the two representative roll angles are presented at the bottom of the figure. As indicated by these curves, the various distributions for a given Mach number may differ considerably. Obviously, this relationship reduces to the transonic area rule at a Mach number of 1.0.

Hayes relates the disturbances in the field to the source and sinks over the surface of bodies in Mach planes similar to those described above. By adopting a simplified relationship between the flow strength and the geometry of the bodies, namely the flows strength is proportional to the normal component of the stream velocity at the body surface, it is possible to utilize Haye's formula to determine the wave drag in terms of area distribution. The theoretical development of the supersonic area rule on the basis of Haye's theory has been done by Robert T. Jones of the Ames Research Center (Reference 8) and G.C. Grogan, Jr. of the Fort Worth Division of Convair.

As a means of visualizing this relationship of wave drag with axial developments of cross-sectional area, equivalent bodies of revolution similar to those utilized in describing the transonic concept could be developed from these axial developments of cross-sectional area. However, it should be emphasized that these bodies have no physical significance since the flow fields produced by each of the bodies is not the same as the flow field produced by the wing-body combination as described for the Mach number of 1.0 case. A small portion of the flow field for a wing-body combination may be approximately the same as that for each of the equivalent bodies.

The application of the supersonic area rule is usually simplified without a significant loss in effectiveness by considering only the cross-sectional areas of the fuselage intercepted by normal cuts. The application is further simplified by using for wing cross-sectional areas the areas of sections normal to the plane of the wing through the intersection of the Mach planes with the plane of the wing. Analysis of area rule computation and fuselage designs made with varying numbers of area developments has indicated that usually the results obtained by using more than three area developments should not result in significant improvements beyond those obtained with three.

The wave drag for the wing-body combination may be computed by determining the wave drag for a group of distributions of cross-sectional area intercepted by the Mach planes utilizing von Karman's well-known formula (Reference 9). Numerous such computations of the wave drag have been made by members of the staffs of the NACA and many aircraft manufacturers. In general, the agreement between theory and experiment is relatively good particularly in defining the effects of the minor modifications in the aircraft configuration. Some of the comparisons of theory and experiment are presented in figure 9. The experimental wave-drag coefficients are plotted vertically and the calculated wave-drag coefficients plotted horizontally for Mach numbers of 1.3 and 1.9. (The flagged symbols are for Mach number of 1.9, the plain symbols for 1.3). The calculations are unsatisfactory near a Mach number of 1.0, but improve rapidly as a Mach number of 1.3 is approached. The 45° line represents perfect agreement. The agreement in many cases is good; however, the worst agreement is of the order of 20 percent. This should be expected since the area rule is a first-order effect and does not include such effects as flow separation and local flow fields produced by individual components.

Fuselage Shaping

On the basis of this concept, the approximately minimum wave drag for a wing-body combination at a given supersonic speed would be obtained by shaping the body so that the various area distributions for this speed approach those for bodies of revolution with low wave drag. Lomax and Heaslet

(Reference 10) have found that the minimum drag obtained by an axially symmetric indentation of the fuselage is obtained by subtracting the average of the area developments for the wing from the minimum drag body for the fixed conditions under consideration. In general, for a combination of a practical wing with a circularly indented body, the area distributions for the various values of θ will deviate from the most desirable shapes. The possibilities of improving the various area distributions at and off the design conditions through the use of body indentations are strongly dependent on the geometry of the wing, as will be discussed more fully later.

Investigations have been made of a number of axially symmetric fuselage modifications designed on the basis of the supersonic area rule to reduce drag at low supersonic speeds. Results obtained for one such experiment are presented in figure 10. The wing of the configuration has 45° of sweepback of the quarterchord, aspect ratio of 4, taper ratio of 0.15, 65A206 section at the root, 65A203 section at the 50-percent semispan, and 65A203 sections at the root and tip. The fuselage has been shaped circularly by subtracting the average area development for three rotations of the cutting plane from a Sears-Haack area distribution for minimum drag with a given volume. The incremental drag coefficient with reference to the drag coefficient obtained at a Mach number of 0.8 are presented. The results indicate that an indentation designed on the basis of the subsonic area rule for a Mach number of 1.2 provides significantly less wave drag than the indentation designed for a Mach number of 1.0 at Mach numbers greater than 1.1. An indentation designed in a similar manner for a Mach number of 1.4 results in an increase in drag over that for the 1.2 indentation at all test Mach numbers. However, the slopes of the curves are such as to indicate that at Mach numbers greater than 1.4, the 1.4 indentation should provide lower drag than the 1.2 indentation. These results together with numerous other results indicate the advantage of designing supersonic airplanes on the basis of the more complex supersonic area rule rather than the simple sonic area rule. The problem of selecting the optimum design Mach number for a given airplane design will be considered later.

It would be expected that greater drag reductions could be obtained by shaping the fuselage non-axially-symmetrically so as to provide improved individual area developments for the various rotations of the cutting plane. The problem of actually determining theoretically the best non-circular shape for the fuselage cutout for any wing at any specified Mach number has been undertaken by Harvard Lomax and Max A. Heaslet at Ames Research Center (Reference 10). By admitting singularities of higher order-quadrupoles, etc., which would distort the rotational symmetry of the fuselage, Lomax and Heaslet have been able to show that the wave drag of a given wing-body system can be reduced, in principle at least, to a minimum value associated with the given overall length and volume of the system, that is, to the value for a simple Sears-Haack body containing the whole volume of the system. This value is, of course, not an absolute minimum for a given volume since, as shown by Ferrari, the wave drag of a body can be reduced to zero by special volume distributions. The resulting body contains two types of distortion. One type is the axisymmetric distortion due to the sources. The other type is the nonaxisymmetric distortion due to higher order multipoles. Figure 11 shows the experimental verification of the ability of these distortions to produce drag reductions. The theory was applied to a wing of elliptic plan form for a design Mach number of 2. Near the wing, the body has a dimpled shape. In the experiments, transition was fixed to minimize change in viscous effects. The quantity C_D is the drag with the distorted body minus the drag with the undistorted body, so that negative values represent drag reductions. The theoretical drag reduction at the design Mach number is shown. A significant portion of this predicted reduction is realized experimentally over the Mach number range of 1.1 to 1.4.

Moment-of-Area Rule

Baldwin has expressed the wave-drag equation for the supersonic area rule in powers of the speed parameter shown at the top of figure 12. The constant coefficients are independent of Mach number and depend only upon the configuration geometry, that is, upon distributions of area and moments of area about the longitudinal axis. In general, a_0 depends only upon the area distribution, a_2 depends upon the second-moment-of-area distribution as well as the area distribution, a_4 depends upon the fourth- and second-moment-of-area distributions and the area distribution, and so on. For a Mach number of 1, the drag equation becomes a function of the area distribution only, and thus reduces to that of the transonic area rule. As the Mach number is increased above 1 the drag becomes dependent upon the distributions of the second and higher order moments of area. The theory thus offers, in principle at least, a means of optimizing the geometry of a configuration at a Mach number of 1 in order to obtain a low wave drag at that Mach number and to obtain a low rate of increase in drag as the Mach number is increased above 1. In the applications of the moment-of-area rule made thus far, however, low drag has been obtained for only sonic and low supersonic speeds, as only the distributions of area and second moment of area have been optimized.

As an illustration of the application of the optimization procedure, consider the wing-body combination shown on the left in figure 12. Also shown directly below, in solid lines, are the distributions of area and second moment of area for this basic configuration. The second-moment-of-area distribution of the body is small compared with that of the wing and is therefore neglected. The shapes of these distribution curves are not conducive to low drag in that the area distribution has a bump at the location of the wing and the second-moment-of-area distribution is short and has steep slopes. For a given volume and length the optimum shapes of the distribution curves are shown by the dashed lines. The optimum second-moment distribution is obtained by utilizing auxiliary bodies of revolution, or pods, mounted on the wing as shown on the right in figure 12. The optimum area distribution is obtained by reshaping the body after the pods have been added. Experimental values of the zero-lift wave-drag coefficients for both the basic and the modified configurations are shown in figure 13. Also shown for comparison are experimental values of the wave drag of a similar configuration modified according to the transonic area rule. Note that the moment-of-area-rule modification resulted in lowest wave drag at all Mach numbers. The higher drag for the transonic area-rule modification at a Mach number of 1 is believed to result from effects associated with the greater slopes of the body indentation on that configuration.

CONSIDERATIONS REGARDING THE APPLICATION OF THE AREA RULE TO THE DESIGN OF AIRPLANE FUSELAGES

In designing a fuselage for a practical airplane configuration, many factors other than the simple geometric shape are involved. Consideration must be given to the influences of asymmetries of real airplanes, engine air intake and exhaust flow, fixed areas of the fuselage, and wing parameters and design Mach number. A number of special extensions and techniques have been developed to account for such factors. These considerations do not invalidate the basic idea of the supersonic area rule as previously published but allow a more effective application of this relation to the design of fuselage contours of practical aircraft. No rigorous theoretical justifications are presently available for these considerations, which, for the most part, are based on physical reasoning. However, limited experiments indicate that usually these considerations provide improved drag characteristics. In one, the use of these procedures resulted in roughly a 100-percent increase in the effectiveness of the fuselage modifications.

Effects of Asymmetries

The principal effect not accounted for by the supersonic area rule is that of reflected waves. Thus, in the supersonic-area-rule approximation, it is assumed that disturbances emanating from the wing or fuselage are not influenced by the presence of the wing, fuselage , or tail. In reality, some of the disturbances are reflected by these components. The supersonic area rule would be improved if procedures for accounting for these neglected effects could be incorporated into the rule.

In this section, a simple method is presented for estimating the effect of reflections produced by the wing and tail. Generally, the reflections produced by the body are extremely complex, and no simple method has been developed for handling these effects. However, the influence of these reflections is usually small compared with those produced by the wing and tail.

The problem of reflections of disturbances by the wing or horizontal tail for asymmetrical configurations is illustrated by the sketch in figure 14. Shown is the side view of a symmetric wing in combination with a fuselage indented only above the wing. Most of the disturbances from the indentation above the wing which are directed downward are reflected upward by the wing as shown; thus, the body shaping above the wing should have little effect on the flow below the wing and an exaggerated effect above the wing. (For symmetrical configurations, the reflection of disturbances produced by changes in the fuselage shape below the wing replaces the disturbances produced by the upper part which could not pass through the wing. For such configurations, the reflection effects are accounted for by the basic area rule.)

The adverse effects that may be associated with such reflections of disturbances by the wing for asymmetrical configurations are illustrated by the zero-lift drag results presented in figure 15. A delta wing having symmetrical airfoil section was investigated in combination with an unindented fuselage and two indented fuselages. In one case, the normal cross-sectional areas of the wing were removed axially symmetrically from the fuselage. In the other case, the total wing cross-sectional areas were removed only above the wing. With the asymmetrical indentation, the incremental drag coefficient C_{D_o}, which is based on fuselage frontal area, was considerably higher than that for the symmetrical indentation throughout the Mach number range of the test. Further, the asymmetric indentation produced adverse effects on the drag compared with those obtained with no indentation at Mach numbers above 1.1. Similar adverse effects would be expected for an airplane configuration with symmetrical wing sections with fuselage shaping concentrated above or below the wing. For lifting conditions, an asymmetric fuselage of the type shown in figure 15 may result in reductions in wave drag.

For the usual design conditions, the problem of determining these reflected effects exactly is extremely complex since the reflection is only partial. However, a reasonable approximation of the effect is obtained by assuming that the reflection is complete for disturbances originating in the region of the wing and is not present for disturbances produced by the fuselage ahead of and behind the wing root. With such an assumption, the areas of the fuselage above and below the plane of the wing are considered separately; while ahead of and behind the wing, the complete fuselage areas are utilized. Such a procedure is strictly applicable only when the wing leading edge is supersonic. However, experimental results for several asymmetrical configurations, including those presented in figure 15 have indicated that fuselage contours based on these separate area developments provide increased reductions in drag even at lower Mach numbers. The areas above and below the horizontal tail are separated in a similar manner.

The cross-sectional areas to account for the reflected disturbances above or below the wing or tail plane can be estimated by the method of images. Within the simplifying assumption described earlier, the flow in the region above the wing plane is the same as that for a wing fuselage composed of the area development above the wing plane plus its mirrored image. Then the effects of the reflected waves are estimated by considering a configuration having twice the areas of the wing and fuselage above the wing plane. A corresponding procedure is utilized to obtain the areas for below the wing plane.

In general, the combining of the area developments above or below the wing or tail planes with the complete areas for the fuselage ahead or behind the wing or tail to form complete area developments will result in discontinuous developments. These discontinuities do not represent real effects of the asymmetric configuration on the flow. Since the wave drag is related to the longitudinal rate of change of cross-sectional area rather than to the cross-sectional area, the real effect is approximated by shifting the areas, so that area developments are continuous. Representative area developments for above and below the wing plane as determined by this method are shown in figure 14.

Inlet and Exhaust

Several experimental investigations have demonstrated that the equivalent stream tube area for the air swallowed by the engine air inlet should be subtracted from the total area development for the configuration from the air inlet to the exit to obtain the most satisfactory interpretation of the wave drag using area development. For most airplanes, the general stream flow usually separates from the fuselage surface at the corner of the jet exit. Beyond this station jet and separated flow displace the unseparated stream. The general displacement of the stream generally expands downstream. For configurations with nacelle-mounted engines, displacement of the jet may have a significant influence on the drag. For such cases, such an expansion of the jet should be taken into account. However, for the calculation of the wave drag for configurations with bases which form the end of the effective area development, the addition of constant area at the base would seem satisfactory for the present. This area is obtained by subtracting the stream-tube area entering the inlet from the base area.

Longitudinal Shape of Fuselage

For most practical airplane configurations, the cross-sectional areas are usually fixed near the nose, in the midregion, and near the tail as shown in figure 16. Obviously, in the midregion, the fixed area includes the average wing areas superimposed on fixed fuselage areas, whereas near the rearward end of the airplane, the average tail areas are added to the fuselage area. At supersonic Mach numbers for the fineness ratios utilized for practical aircraft, the slender-body theory does not provide a reliable indication of the area development for minimum wave drag for such fixed conditions; and, the nonslender linear theory should be utilized. However, because of the extreme complexity of this more inclusive theory, its use for the computation of the exact minimum-drag developments for these fixed conditions is impractical at present. Parker (reference 11) has obtained a nonslender solution for the conditions of fixed lengths and fixed maximum areas at the mid-body station. Approximations of the developments for the fixed conditions of practical airplanes may be obtained through a consideration of the developments for these simpler conditions. Although such developments depend on fineness ratio and Mach number, for the values of fineness ratio of practical interest and for Mach numbers from 1.2 to 2 (the probable range in which fuselage contours will be designed on the basis of the area rule) the shapes are approximately the same. The shape for a mean condition of these ranges is shown in figure 16. This shape is based on an interpolation of shapes obtained by Parker. It may be noted that this development has a corner and consists of approximately straight lines over most of the length. This suggests that the minimum-wave-drag envelope for the fixed conditions shown in figure 16 might be approximated by fairing straight lines tangent to the fixed area developments as shown.

The effects of fuselage modifications based on several envelope area developments (figure 17) have been determined during the tests of an airplane with 42° swept wings. One envelope utilizes the straight-line fairings based on nonslender body theory as just discussed; the other approximates that which would have minimum wave drag based on slender-body theory. The design Mach number was 1.2. The incremental minimum wave-drag coefficients which are based on wing area, for Mach numbers to 1.2 are presented in figure 17. The drag results indicate that the fuselage contours based on the straight-line-fairing envelope produce significantly lower wave drag than did the contours based on the approximation of the slender-body theory. Before the nonslender-body theory became available, the use of the straight-line envelope rather than the slender-body-theory minimum-drag envelope was proposed as a means for improving the drag at off-design conditions. Experimental results for several configurations indicate that the use of such an envelope in preference to one based on slender-body theory reduces the wave drag at off-design conditions considerably more than at the design conditions.

Cross-Sectional Shape

The cross-sectional shapes for the fuselage arrived at by the more comprehensive theory of Lomax and Heaslet (reference 10) may be impractical. However, the results of these calculations suggest practical shapes which should provide improved reductions of wave drag. When feasible, such changes in shape should be concentrated on the sides of fuselage. When the depth of the fuselage side above or below the wing or tail is relatively small, such a procedure may result in excessive changes of the slopes of the lines of the fuselage sides, undesired increases in the fuselage wetted area, and unwieldy distributions of fuselage volume. For such conditions, the required changes in the fuselage area development have been distributed around the top or bottom of the fuselage as well as on the sides. Where possible, the changes in the fuselage area development intended to offset disturbances produced by canopies, stores, nacelles, fairings, and other similar components producing changes in the area developments should be placed as close as possible to these components.

Wing Configuration and Design Mach Number

The range and relative magnitude of the favorable effects of body shaping based on the supersonic area rule are markedly influenced by the wing configuration. Comparisons of a number of experimental results have indicated that the general overall effectiveness of body shaping is usually greater with increased wing or tail leading-edge sweep. Also, comparisons of unpublished experimental results have shown that the relative effectiveness of body shaping is larger with the centroids of the cross-sectional areas of the wing or tail closer to the fuselage. Such inward positions of these centroids are generally associated with lower aspect ratios and taper ratios of the plan forms and the use of spanwise reduction in the section thickness ratios from root to tip. Greater wing sweep or inward positions of the centroids generally causes the area distributions for the wing and tail surfaces for the various values of θ at the various Mach numbers to approach more nearly the average distribution used to design the fuselage contours. Thus, the irregularities in the area distributions for these various conditions are less and the wave drag is less.

The design Mach number for the fuselage contour which would provide the optimum compromise performance for the range of operation of an airplane is dependent on the relative importance of operations at the various conditions and the effectiveness of the shaping at these conditions. Unfortunately, little information is available on the effectiveness of various body shapes for wide ranges of conditions. Until such information is obtained, it would seem advisable to design the contour for a specific Mach number equal to a weighted average operational speed, within the limitations discussed below.

Comparisons of area developments and limited experimental results have indicated that, when a fuselage is shaped on the basis of the supersonic area rule for a Mach number significantly greater than that for which the leading edge becomes supersonic, the reduction in drag at the design condition is generally only slightly greater than that obtained at this speed with a shaping designed for a Mach number less than the critical value. On the other hand, the reductions in drag at Mach numbers below this critical value obtained with such a shaping are considerably less than those resulting from shapings designed for these lower speeds, particularly at Mach numbers near 1.0. Therefore, the fuselage should generally not be designed for a Mach number significantly greater than this critical value, even though the average operating speed of the airplane may be well beyond this speed.

DELAY OF DRAG RISE

In this section, consideration is given to some recent results obtained on modifications to the fuselage and wing based on the area rule intended to delay the onset of the drag rise at high subsonic speeds. Configurations with thicker, more highly cambered, high aspect ratio wings generally utilized for airplanes designed to cruise at high subsonic speeds are considered. At cruise conditions, such airplanes usually operate at moderate lift coefficients; therefore, the considerations will be limited to such conditions. For such wings operating at moderate lift coefficients the initial drag rise is usually associated with shock-induced separation on the upper surface of the wing. This separation normally not only causes drag rise but also produces variations in lift, adverse changes in the pitching-moment characteristics, and buffet. The modifications to be discussed are intended to delay the initial onset of the shock wave, thus delaying the associated boundary-layer separation and the resulting adverse effects. Such modifications should normally provide directly an increase in the Mach number for drag rise of roughly .07. With the increase in wing sweep which such modifications make practical the delay of drag rise should be roughly 0.14 Mach number.

Streamline Fuselage Shaping

Fuselage modifications to delay the onset of the initial shock wave of the sweptback wings were proposed before the discovery of the area rule. Such modifications were intended to provide an alignment of the fuselage surface with the streamlines over an infinite sweptback wing, thus reducing the adverse interference of the fuselage on the local flow field over the root sections of the wing. This approach was originally proposed by Küchemann in Germany; however, it was also independently proposed by Watkins of the Langley Research Center. A number of experiments were made of such fuselage modifications. Representative results of one of the more recent investigations of such modifications made in the Ames 16-foot high-speed wind tunnel are presented in figure 18 (reference 12). The wing of the configuration had 35° of sweepback, an aspect ratio of 6, taper ratio 0.5, and NACA 64A015 sections normal to the 50-percent chord line. The basic bodies of the configuration were shaped according to Sears-Haack formula for minimum drag with a given volume. The body was modified around the entire periphery in an attempt to provide a flow at the root of the wing essentially the same as for infinite span wing. The theory utilized in this design has been developed by McDevitt as an extension of the theory proposed by Küchemann and Weber presented in reference 13. The modification consists essentially of an indentation. The results presented in the figure are for a lift coefficient of 0.3. It may be seen that this body shaping provided a slight delay in the drag rise Mach number and a considerable reduction of the drag increase at Mach numbers of the order of 0.9.

Area Rule Fuselage Shaping

The results obtained with Mach number of 1 area-rule fuselage shaping have usually indicated that such shapings provide not only a reduction in drag at Mach number of 1, but also a significant delay of the drag rise Mach number both at high lift and lift conditions. This delay is usually of the same order as the delay provided by the streamline shaping just discussed. More recently, investigations have been made of special versions of these area-rule shapings intended to provide greater applicability of application as well as improvements of effectiveness in delaying drag rise Mach number (Reference 14). For existing airplanes and for designs where the minimum fuselage dimensions are established by clearance requirements, fuselage shapings intended to delay the drag rise usually would be accomplished by increasing the volume of the fuselage or by attaching appendages to the primary fuselage structure. The practicability of the application of such fuselage additions is generally increased by concentrating such additions on limited regions of the fuselage, inasmuch as this procedure generally results in a simplification and a reduction of weight of the fuselage structure and an increase in the usability of the added fuselage volume. Emphasis, therefore, has been placed on such concentrated additions in the present study.

The onset of drag rise for an airplane with a relatively thick wing at cruise lift coefficients is usually caused primarily by boundary-layer separation on the upper surface of the wing resulting from the development of an initial shock wave above the wing. The shock-induced separation is usually less severe on the fuselage and the inboard sections of the wing than on the midsemispan region of the wing. The difference is particularly great for sweptback wings. Fuselages shaped to improve the longitudinal area development for the airplane tend to reduce the strength of the initial shock over the inboard region of the wing, thereby further increasing the difference in extent of separation along the wing span. Increasing the lift on the less critical upper surfaces of the fuselage and the inboard sections of the wing, thereby allowing a decrease in lift on the more critical outboard region, should result in a decrease in boundary-layer separation along the outboard wing surface. It would be expected that the favorable effect of this reduction in separation on the midsemispan region of the upper surface would be considerably greater than any adverse effects of increasing the lift on the fuselage and inboard sections of the wing, and thus an overall improvement should result.

In the present design, the localized increases of lift on the upper surfaces of the fuselage and inboard sections of the wing have been accomplished by incorporating camber in the fuselage. For the usual fuselage additions intended to improve the area developments, with forward and rearward additions above and below the wing, the desired camber is effectively obtained by adding vertically to the upper forward and lower rearward parts of such additions and subtracting vertically from the lower forward and upper rearward parts. Such a modification should also provide a favorable increase in lift on the lower surface of the wing.

Schlieren photographs have indicated that for the speed and lift conditions at which the fuselage shaping normally is most useful, the flow is usually supersonic by a considerable degree in a relatively large local region above the upper surface of the wing. Therefore, it would seem probable that to obtain the greatest reduction in the strength of the initial shock wave in this speed range, the fuselage addition above the wing should be shaped longitudinally to improve the area developments obtained with the oblique cutting planes associated with supersonic fields. Such a shape may be approximated by moving the shaping for a design Mach number of 1.0 somewhat forward.

Fuselage additions shaped on these considerations have been investigated on the configuration shown in figure 19. The wing had 35° of sweepback of the quarter-chord line, an aspect ratio of 7.05 and a taper ratio of 0.38. The wing sections varied linearly from a NACA 65A213, a = 0.5 (approx.) section at the wing-fuselage juncture to an NACA 65A209, a = 0.5 (approx.) section at the 0.38-semispan station, with an NACA 65A209, a = 0.5 (approx.) section from that station to the tip. Neither wing had any built-in twist or dihedral. The basic fuselage utilized in the investigation is cylindrical in the region of the wing.

Inasmuch as the total lower and the reduced upper rearward additions to the fuselage cannot affect to a large degree the boundary-layer separation on the upper surface of the wing, it would be expected that these additions to the fuselage would contribute relatively little to the reduction in total drag rise of the configuration. Therefore, the fuselage addition shown has been limited to the upper forward portion of the fuselage. The addition was concentrated on top of the fuselage. The shape of this addition was obtained by shifting a contour designed for a Mach number of 1.0 forward a distance of roughly 15 percent of the wing-fuselage-juncture chord.

The fuselage addition provides a delay of the drag rise of approximately 0.02 for a lift coefficient of 0.3 (figure 20). The drag rise Mach number has been arbitrarily chosen as the value at which $C_D/M = 0.10$. The delays for lower and higher lift coefficients are less than for a lift coefficient of 0.3. The drag benefits associated with the proposed method appear to be of the same order as those obtainable with the Küchemann "streamline" method for a comparable configuration. (See reference 12 for example). However, the practicability of application of the present concentrated addition normally should be considerably greater than that of an addition required to provide the Küchemann shaping.

Additions on Wing

The special fuselage addition just described in reference 14 provides a practical means of obtaining a significant delay of the separation on the inboard section of a wing;

however, this addition causes only secondary reductions of separation on the outboard sections. As a means of reducing the separation on these outer sections, a series of special bodies added on the upper surface of the wing are proposed (Reference 15). The forward portions of these bodies decelerate the supersonic flow ahead of the shock wave above the wing with a resulting decrease of the strength of the shock wave and the associated separation.

Investigations of the wing additions have been made using essentially the wing-fuselage configuration shown in figure 19. The added bodies investigated are also shown in figure 19. Four bodies were placed on each wing panel with the body center lines at the 27-, 45-, 63-, and 80-percent semispan stations. The lengths of the bodies were equal to the local wing chord, the noses of the bodies were at 40 percent wing chord, and the maximum body radii were at 90 percent wing chord. The additions were closed bodies of revolution with a fineness ratio of 6.5. At the trailing edge of the wing, 90 percent of the body diameter was above the wing. Rearward of the trailing edge the lower part of the body of revolution was removed so that none of the body extended below the wing chord plane.

Since the spanwise extent of the favorable effects of the added bodies on the flow field above the wing is relatively limited at Mach numbers under consideration, a number of such bodies must be utilized to obtain a satisfactory reduction of separation across the span. For configurations with fuselage additions similar to that utilized in the present investigation, no bodies need be placed on the wing near the fuselage. The surface flow surveys presented herein indicate that with such additions, the shock-induced boundary-layer separation is effectively eliminated near the fuselage at the conditions under consideration.

The longitudinal area developments of cross-sectional area for the added bodies have been arrived at using a special extension of the area rule. At the conditions under consideration, the local shock wave above the wing and the flow field causing the wave, while not as broad as those present at Mach numbers near 1.0, do have considerable vertical extent (see reference 16). An indication of the general influence of the added bodies on this moderately extensive field should be provided by an analysis of the effects of the cross-sectional areas of the bodies on the area developments for localized chordwise segments along the span of the wing. The analysis should be most indicative with a wing of high aspect ratio and the added bodies relatively closely spaced, as for the configuration of the present investigation. The most readily interpreted results are obtained by considering segments which include single bodies. Because the changes in the flow field above the wing at low lift coefficients are influenced primarily by the shape of the upper surface, only the cross-sectional areas for above the chord plane have been considered. Further, since the effects of the portions of the bodies aft of the trailing edge are secondary and relatively complex, the cross-sectional area developments for these parts are not included in the analysis.

For a swept wing, similar to that of the present investigation, the area-rule analysis is further modified. For such a wing having a high aspect ratio, with a shaped fuselage, the flow about the critical midpanel region approximates that for an infinite span swept section. Since the flow about such a section is primarily dependent on the shape of the section perpendicular to the swept elements (reference 17), it would seem reasonable that the flow about the wing segments would be most closely defined by developments of the cross-sectional area obtained along the swept elements. With the added bodies relatively closely spaced, an approximate indication of the general effect of the added bodies on the flow over the swept segments should be obtained by adding the oblique cross-sectional areas of the bodies intersected by the cutting planes through the elements of the wing to the corresponding areas for the wing segments. To simplify the computations, normal components of the oblique areas for the wing segments and normal areas for added bodies have been used in the present analysis. These normal components for the wing have been approximated by multiplying the vertical ordinates of the mean section by the width of the segment.

The shape of the total area development utilized has been arrived at by considering two-dimensional airfoil theory and experimental data, since, with a relatively close spacing of the bodies, the general flow above the wing segments and bodies is probably similar to that above a two-dimensional wing section with a similar upper surface area development. A general correlation of drag rise with airfoil shape suggests that for most usual airfoils with reasonably smooth contours the initial drag rise at low lift coefficients is primarily dependent on the curvature, or ratio of change of slope, of the upper surface near the maximum ordinate of that surface. A reduced curvature results in a delay and reduction of the initial rise. For example, the mean curvature over a 40-percent chord region centered on the maximum thickness for an NACA 16A009 section is approximately 75 percent of that for an NACA 65A-009 section; the initial drag rise Mach number for the 16-series section at a lift coefficient of 0.2 is approximately 0.03 higher (reference 16). An equivalent reduction of the curvature in this region associated with a decrease in thickness ratio results in a similar delay of drag rise for 64-series airfoils (Reference 16). Obviously, the changes in the shapes in the fore and aft portions of the airfoil associated with the decrease in thickness are completely different from the variations between the 16- and 65-

series airfoils in these regions. Therefore, the total area developments for the bodies of the present investigation have been designed primarily to provide a relatively gradual rate of change of slope of the development near the peak area as for 16-series airfoils.

The maximum cross-sectional areas for the bodies near the trailing edge are approximately half of the maximum areas for the wing segments. Experimental results indicate that the use of significantly smaller bodies results in marked reductions of effectiveness in reducing drag rise. Use of larger bodies results in some increase in effectiveness at the higher lift coefficients. However, the value of this improvement is probably less important than the associated adverse increase in skin-friction drag and weight.

Tests of the configuration were made in the Langley 8-foot transonic tunnel over a Mach number range from 0.60 to 0.95. The tests were conducted at a Reynolds number of approximately 4×10^6 per foot. All configurations were tested with transition fixed by roughness strips at the 10-percent-chord stations on the upper and lower surfaces of the wing. The strips were 0.1-inch wide and consisted of No.120 carborundum grains with approximately 20 grains per inch.

The lessening of separation associated with adding the bodies to the configuration with a fuselage addition resulted in substantial reductions in drag at lifting conditions for Mach numbers greater than approximately 0.86. For a lift coefficient of 0.3 at a Mach number of 0.92, the drag coefficient was reduced by approximately 0.012 (figure 21). These reductions in drag provide significant delays of drag-rise Mach number. At a lift coefficient of 0.3, the Mach number at which $C_D/M = 0.1$ is increased by approximately 0.05 (from 0.85 to 0.90). At speeds just below the abrupt drag rise the drag increases slightly, because of the local regions of separation and boundary-layer thickening.

Delays of shock-induced boundary-layer separation can, of course, also be obtained practically by lengthening the chord of the wing, thus providing thinner sections and lower lift coefficients. A chord extension of about 25 percent would increase the low speed level approximately the same amounts as do wing additions. Analysis of wing section data (reference 16, for example) and results regarding the effects of aspect ratio for swept wings indicates that a proportional lengthening of the chord by this amount would delay the drag rise approximately 0.01 Mach number.

The added bodies caused a marked reduction in the loss of lift for a given angle of attack experienced by the configuration without the additions at Mach numbers greater than approximately 0.86. At the lower speeds, the additions resulted in a significant decrease of lift at a given angle of attack. This reduction resulted in no appreciable increase in drag due to lift at these speeds.

Adding the bodies to the wing caused a marked lessening of the non-linearities of the variations of pitching moment with lift for all test Mach numbers (figure 22). Because of the pronounced favorable effect of the added bodies on the pitching-moment characteristics, incorporation of these bodies should allow a significant increase in the wing sweep that can be practically utilized. Such an increase in sweep would provide a further delay of drag rise.

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WING-BODY COMBINATION AND EQUIVALENT BODY OF REVOLUTION

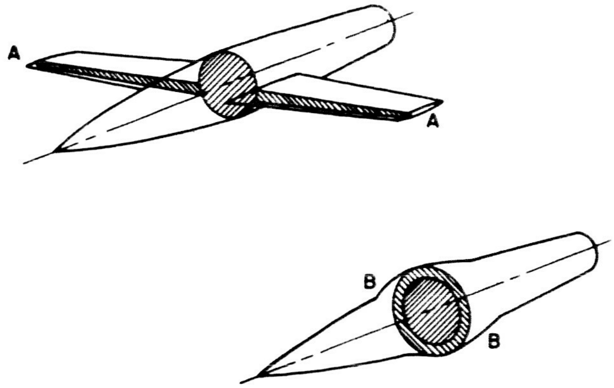


Figure 1.

TRANSONIC FLOW PAST BODY WITH STRAIGHT WING

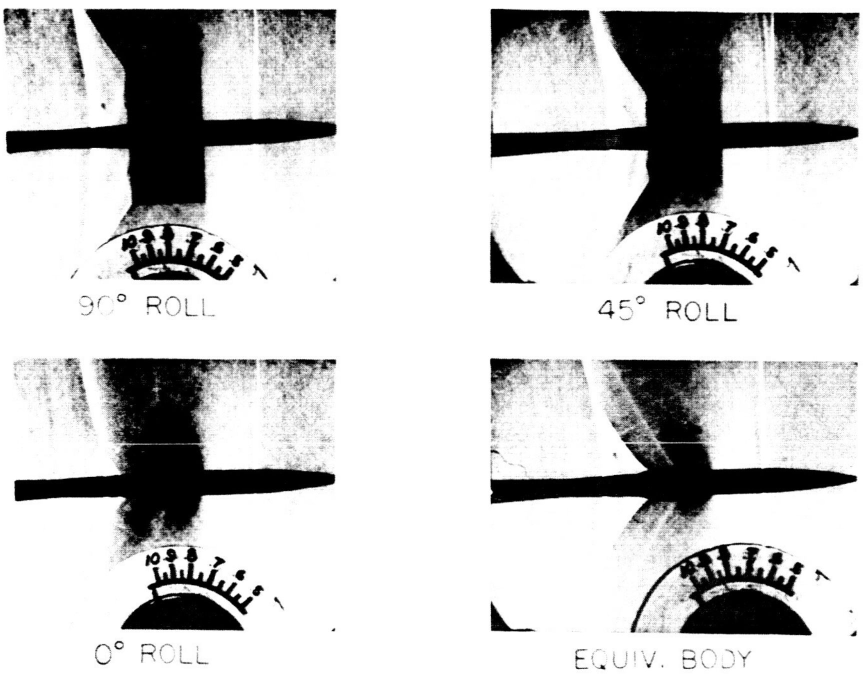


Figure 2.

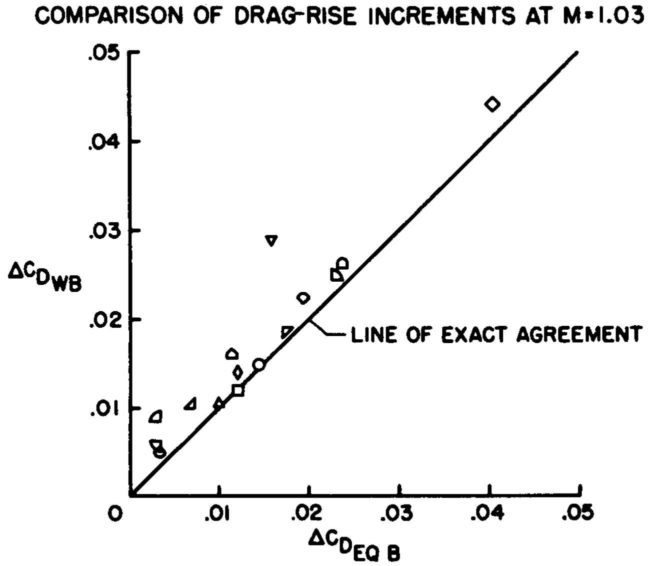


Figure 3.

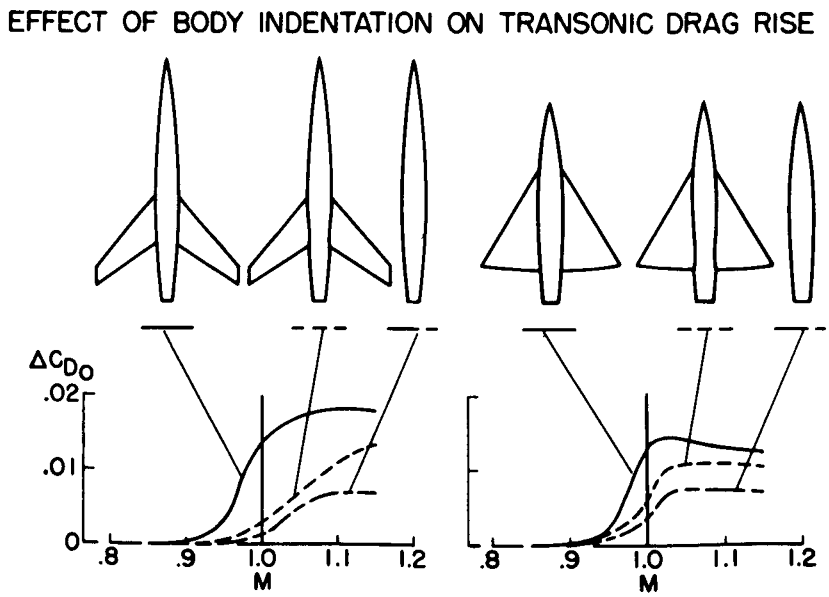
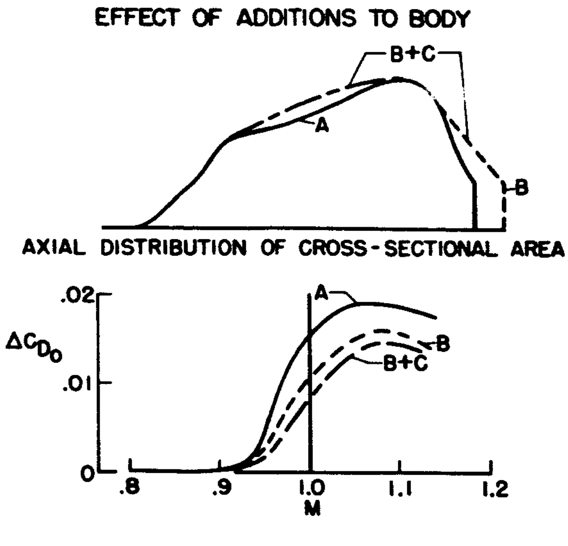


Figure 4.



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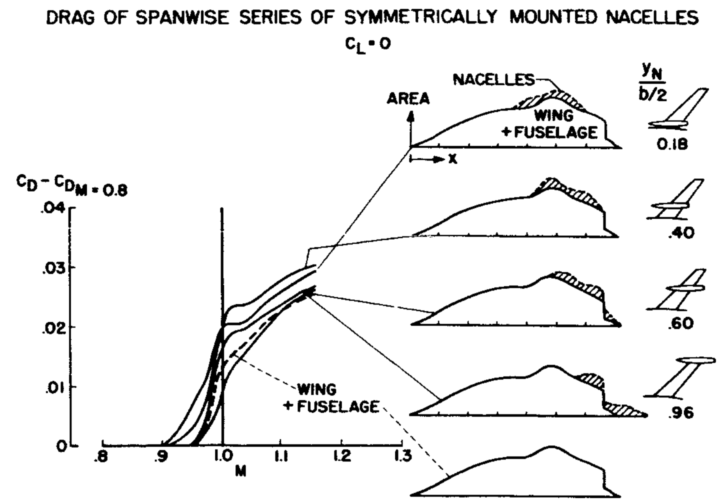

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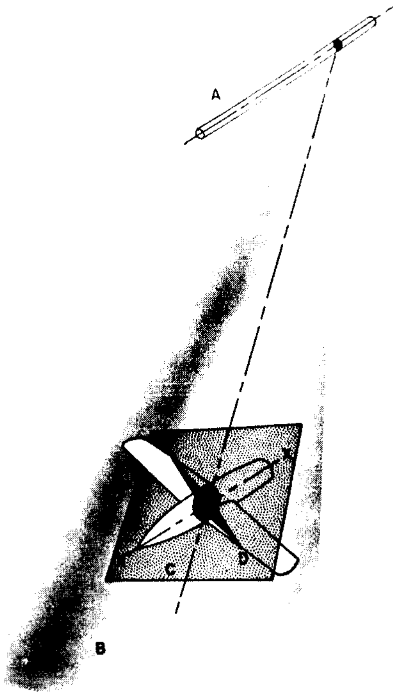

Figure 7.

PROCEDURE USED IN DETERMINING AREA DISTRIBUTIONS
FOR SUPERSONIC SPEEDS

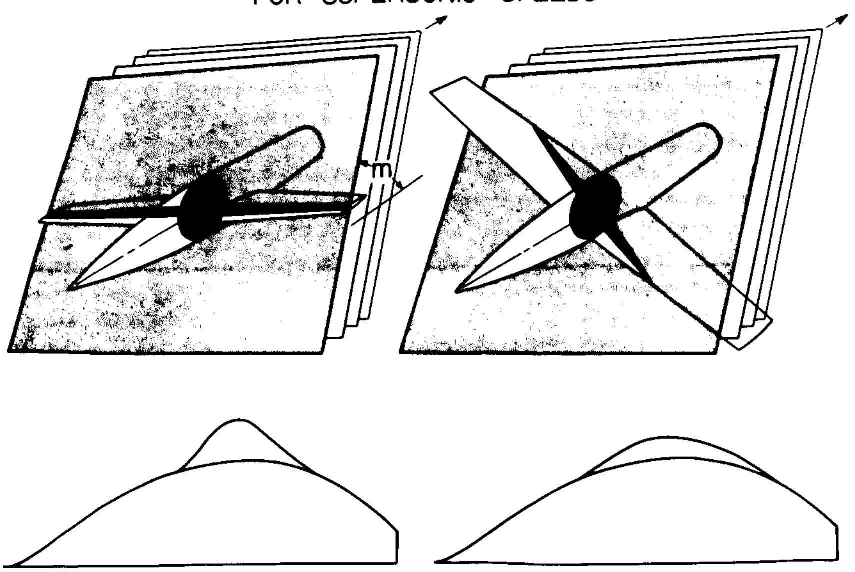

Figure 8.

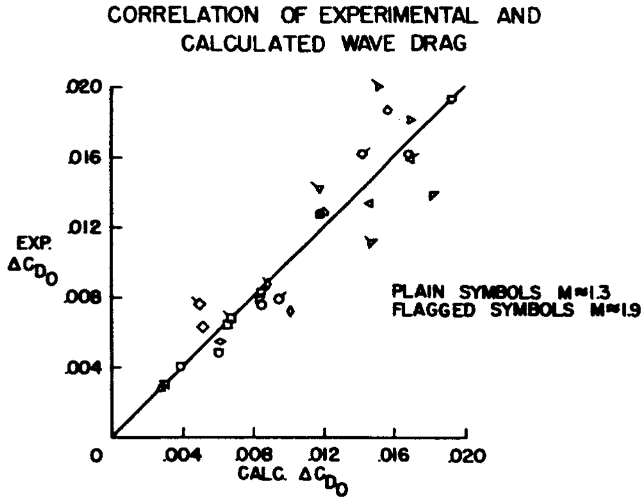

Figure 9.

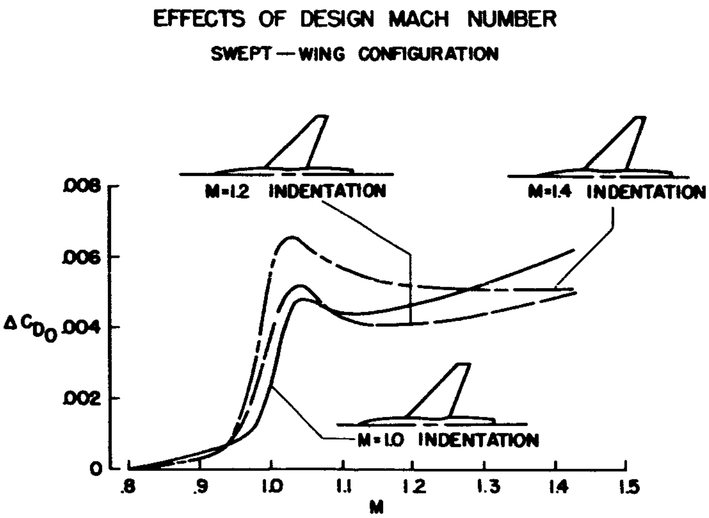

Figure 10.

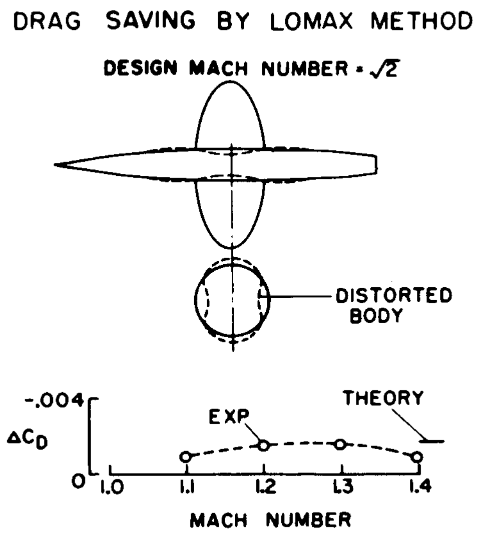

Figure 11.

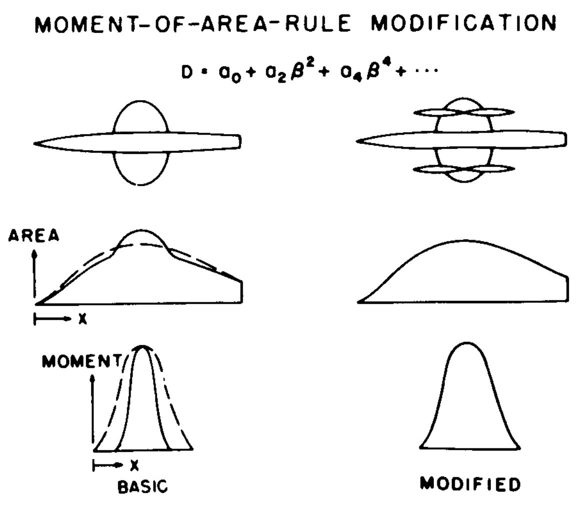

Figure 12.

EFFECT OF MODIFICATIONS ON EXPERIMENTAL WAVE DRAG; $C_L = 0$

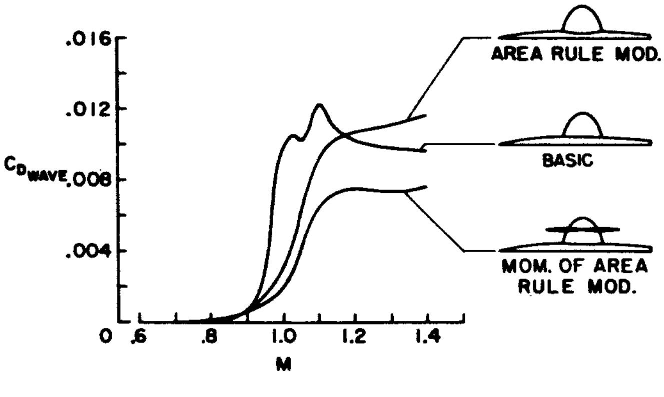

Figure 13.

REFLECTION OF DISTURBANCES BY WING FOR ASYMMETRIC CONFIGURATION

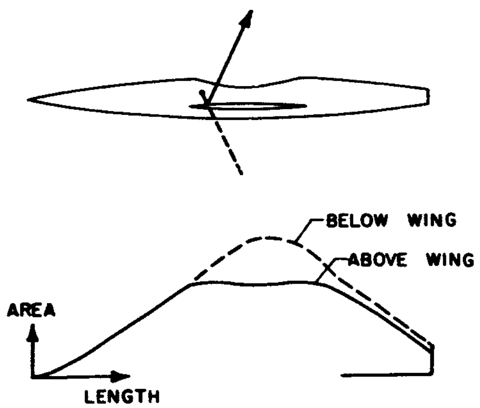

Figure 14.

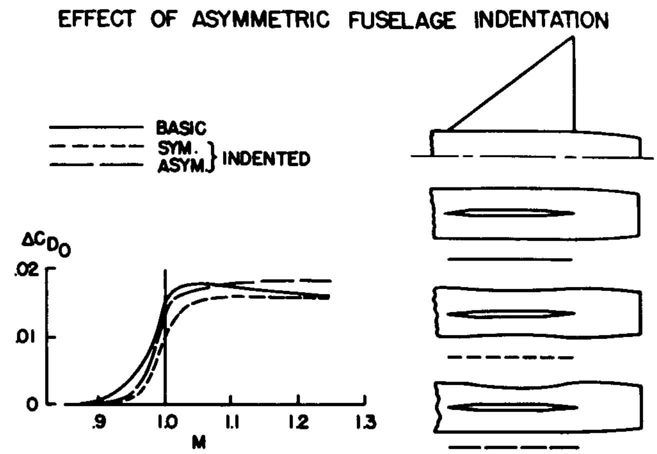

Figure 15.

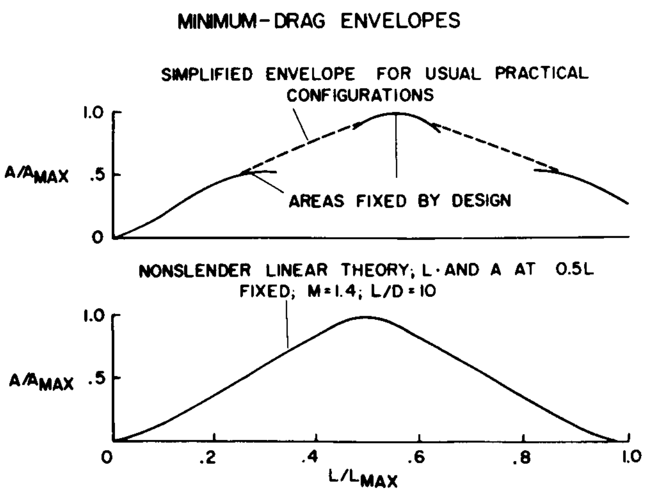

Figure 16.

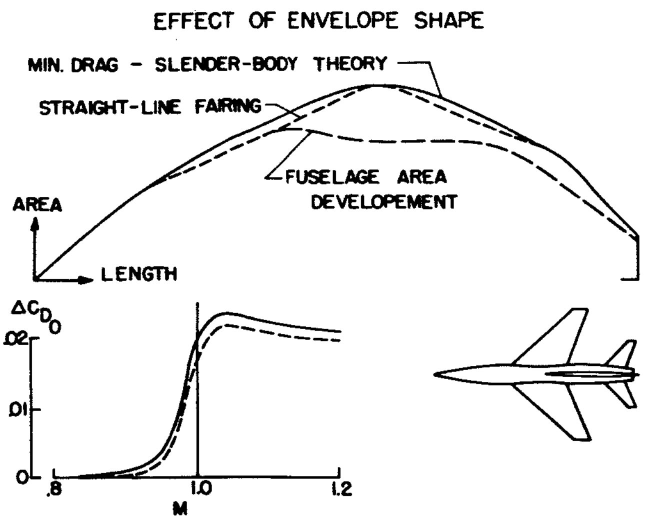

Figure 17.

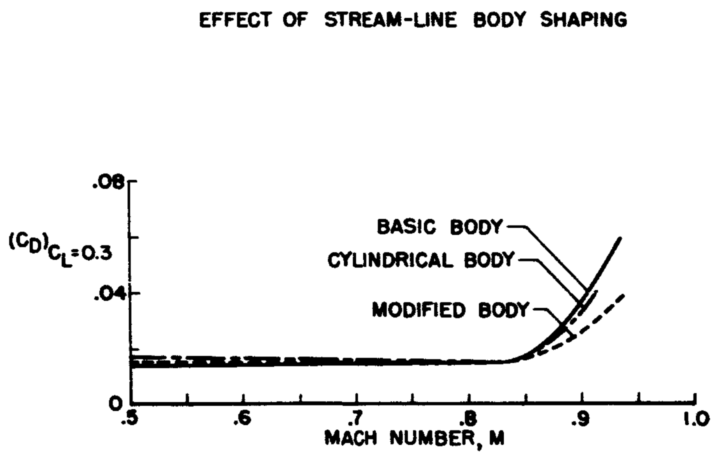

Figure 18.

EXPERIMENTAL CONFIGURATION

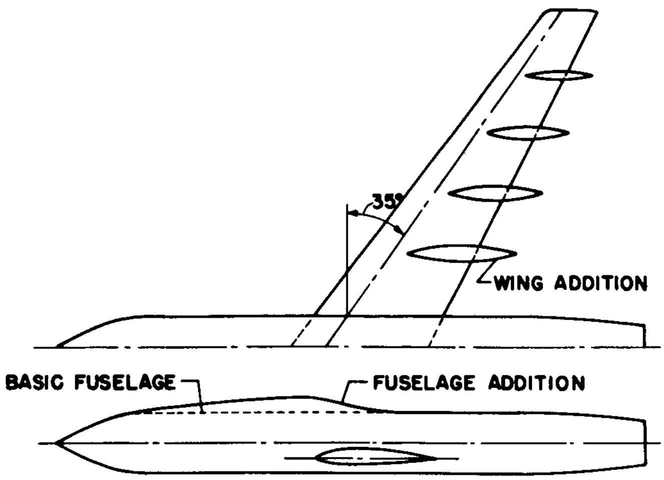

Figure 19.

EFFECT OF FUSELAGE ADDITIONS

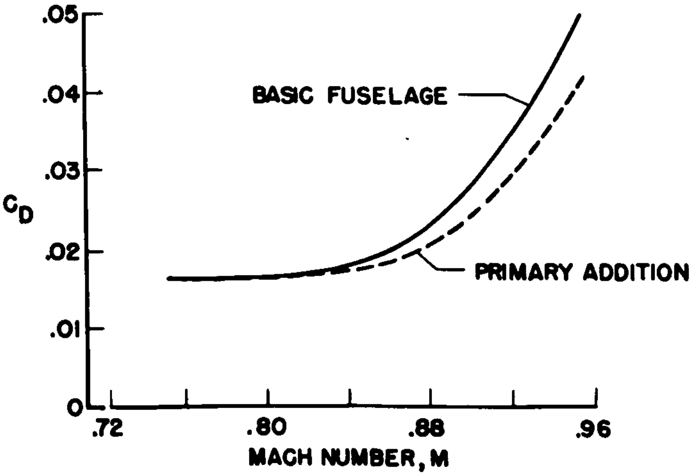

Figure 20.

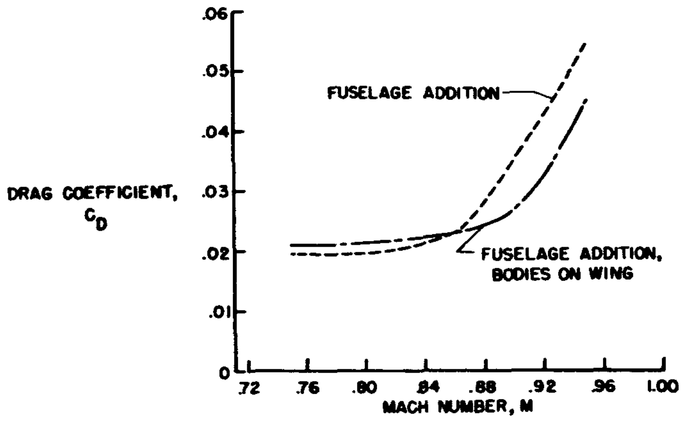

Figure 21.

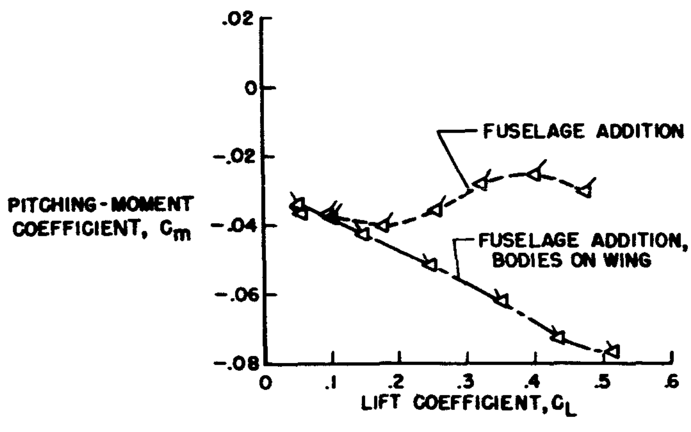

Figure 22.